SmallSat Missions Enabled by Paired Low-Thrust Hybrid Rocket and Low-Power Long-Life Hall Thruster

Ryan W. Conversano
Jet Propulsion Laboratory,
California Institute of Technology
4800 Oak Grove Dr.
Pasadena, CA 91109
818-393-4382
Ryan.W.Conversano@jpl.nasa.gov

Jason Rabinovitch
Jet Propulsion Laboratory,
California Institute of Technology
4800 Oak Grove Dr.
Pasadena, CA 91109
818-354-4713
Jason.Rabinovitch@jpl.nasa.gov

Nathan J. Strange
Jet Propulsion Laboratory,
California Institute of Technology
4800 Oak Grove Dr.
Pasadena, CA 91109
818-203-0726
Nathan.J.Strange@jpl.nasa.gov

Nitin Arora
Jet Propulsion Laboratory,
California Institute of Technology
4800 Oak Grove Dr.
Pasadena, CA 91109
818-354-2417
Nitin.Arora@jpl.nasa.gov

Elizabeth Jens
Jet Propulsion Laboratory,
California Institute of Technology
4800 Oak Grove Dr.
Pasadena, CA 91109
818-393-1003
Elizabeth.Jens@jpl.nasa.gov

Ashley C. Karp
Jet Propulsion Laboratory,
California Institute of Technology
4800 Oak Grove Dr.
Pasadena, CA 91109
818-354-6322
Ashley.C.Karp@jpl.nasa.gov

***Abstract* — The capabilities of a SmallSat-class spacecraft targeting the outer solar system and using a combined chemical and electric propulsion system are explored. The development of compact hybrid rockets has enabled high-thrust engines to be packaged tightly enough to fit on cubesat and SmallSat spacecraft. These hybrid rockets provide 10's-100 N of thrust depending on the propellant load & >300 s of specific impulse and have been demonstrated in both ambient and vacuum environments. Advancements in low-power long-life Hall thruster technologies have provided the potential for significantly greater propellant throughputs, enabling their use as a primary propulsion element on interplanetary spacecraft. In a recent characterization test campaign, the MaSMi-DM Hall thruster demonstrated power throttling from 150 – 1000 W with >1500 s of specific impulse available at >500 W and ≥40% total thrust efficiency available at >300 W; peak values of 1940 s and 53% were observed. A notional low-mass spacecraft employing a combined hybrid rocket and low-power electric propulsion system was designed and used for mission concept analysis targeting the outer solar system. Using an imposed wet mass limit of 400 kg, mission trajectories to Saturn and Uranus were generated. Orbit capture with >40% of the launch mass was shown to be possible at either target, with mission transfer times of 7.5 years and 13.5 years for Saturn and Uranus, respectively. Significant follow-on mission activities near Saturn (e.g. to Titan & Enceladus) were also possible by carrying extra propellant mass while remaining under the total wet mass limit.**

TABLE OF CONTENTS

1. INTRODUCTION

The proliferation of miniaturized spacecraft technologies has enabled the possibility for SmallSat-class (sub-300 kg) spacecraft to perform challenging deep-space missions. Accessing the planets in the outer solar system in reasonable times using SmallSats, however, requires innovative propulsion and power solutions. Recent technology development activities at the Jet Propulsion Laboratory (JPL) has provided an attractive option for low-mass spacecraft to probe the outer solar system: the pairing of a low-thrust chemical rocket with a low-power, long-life electric propulsion (EP) system. This combined propulsion

978-1-5386-6854-2/19/$31.00

architecture would provide a SmallSat the capability to execute time-critical impulsive maneuvers (orbit departure, orbit insertion, etc.) and precise, long duration applications of low thrust (cruise, proximity operations, etc.).

Providing an EP-based spacecraft traveling to the outer solar system with sufficient power is a significant challenge, especially as spacecraft size is reduced to the SmallSat scale. Recent advances in solar array technology has made solar power practical for missions using conventionally sized spacecraft to Jupiter, such as Juno and the proposed Europa Clipper. For many missions to Saturn, nuclear power is a more attractive option because it provides significantly higher specific power than solar arrays. Beyond Saturn, most solar powered spacecraft architectures quickly become infeasible. Next-generation nuclear power options, including radioisotope thermoelectric generators (RTGs) [1] and the potential Kilopower reactor [2], could provide a range of powers that are well suited for the combined propulsion system under investigation. These new nuclear power systems, whether RTGs or reactors, still would have a much lower specific power than solar power in the inner solar system, resulting in a lower thrust to weight compared to solar electric propulsion (SEP) systems. Therefore, this study will focus on mission targets beyond the asteroid belt.

An overview of the propulsion system elements and potential capabilities of an Outer Solar System SmallSat (herein referred to as OSSS, or OS^3) will be the focus of this paper.

2. CHEMICAL (HYBRID) PROPULSION ELEMENT

The chemical element of the combined SmallSat propulsion system is a small-scale hybrid rocket. A hybrid rocket is a type of rocket motor that uses a solid fuel and a gaseous oxidizer. This particular hybrid rocket, which utilizes solid PolyMethylMethAcrylate (PMMA) and gaseous oxygen (GOx), has been under development at JPL for three years [3–5]. The hybrid system has demonstrated high performance with a specific impulse of more than 300 s at a thrust level of ~40 N. As the total propellant load scales up, so would the thrust. The fuel and oxidizer are independently inert and physically separated by both location and phase during storage. From a safety perspective, this simplifies the accommodation and integration of a hybrid propulsion system significantly.

The combination of PMMA/GOx was selected as the ideal propellant/oxidizer combination after a performance evaluation of numerous available options was completed. PMMA, commonly referred to as acrylic, is a thermoplastic that is used for a variety of non-propulsion applications (most commonly a lightweight alternative to glass that is challenging to shatter). Ignoring any potential additives or contaminants, PMMA has the chemical composition of $(C_5O_2H_8)_n$, where $C_5O_2H_8$ is the repeating monomer. This classical diffusion limited hybrid fuel is slower burning which reduces the thrust of the rocket, which is advantageous so that a conventional reaction control system (RCS) system can still be used to control a low-mass spacecraft. A detailed characterization effort of the performance and properties of the PMMA/GOx has been performed as part of the hybrid rocket development effort [3,4].

A methane/oxygen torch is used as the hybrid rocket ignition system. Over a hundred independent spark ignition system tests (i.e. with no motor) have been completed to optimize operating conditions and to ensure reliable and repeatable ignitions. Twenty-four ignitions have been attempted and demonstrated under vacuum. This system was selected to enable multiple restarts of the hybrid motor, a capability which significantly expands the mission space for the technology. The oxidizer for the spark ignition system (also GOx) is shared with the main motor; however, a small additional tank of methane is required. A commercial spark plug is used to ignite the bipropellant mixture inside of the igniter chamber, after which the hot combustion products are fed into the main motor to ignite the solid fuel grain. Details of the efforts characterizing the igniter system can be found in [5].

Hot-fire testing at JPL demonstrating the capabilities of this system consisted of three primary components:

1) PMMA regression rate characterization testing using a subscale heavyweight motor
2) Flight-like motor testing in ambient conditions
3) Flight-like motor ignition testing in vacuum

The subscale motor can accommodate fuel grains ranging is length from 7.6 to 30.5 cm, with a consistent fixed fuel grain outer diameter of 7.6 cm. The flight-like motor is designed to be more mass efficient, and to generate the required delta-V for a real flight mission. It is essentially the flight weight and configuration except for flanges to simplify testing with multiple fuel grains. It can accommodate a ~15.2 cm fuel grain with an outer diameter of 12.7 cm. Only the flight-like motor was tested under near vacuum conditions. Figure 1 shows images of the two different motors being tested under different operational conditions. Regression rate information from the test campaign is available in [4], and application of this regression data to the system-level design can be found in [3]. Two long-duration tests of the subscale heavyweight motor designed to use as close to 100% of the fuel as possible demonstrated a fuel utilization of 94.3% and 97.4% (by mass), with the improvement observed following an improvement in injector design incorporated between the tests. Additionally, a test to confirm end of life performance (a low oxidizer mass flux) showed stable combustion, which indicates that reliable and predictable performance is expected for the hybrid system even when nearly all of its fuel has been expended. All tests performed to date have shown stable and repeatable burns. With the flight-like motor, 24 ignitions were carried out at low pressure (<0.035 mbar). The number demonstrated overall system reliability as no failed ignitions were observed, which allows this propulsion system to support missions that require multiple burns.

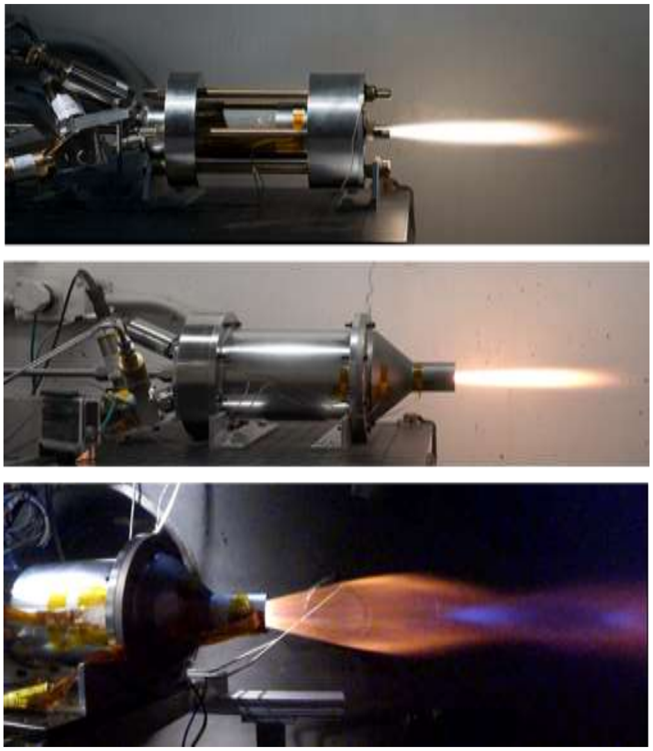

Figure 1. Hotfire tests of the subscale heavyweight hybrid rocket motor in ambient conditions (top), the flight-like hybrid motor in ambient conditions (middle), and the flight-like hybrid motor in vacuum (bottom).

3. Electric Propulsion Element

The EP element of the combined SmallSat propulsion system is a low-power system based on a sub-kW long-life Hall thruster. Numerous commercial Hall thrusters have been demonstrated successfully in space, yet none have been developed and demonstrated to provide both high-efficiency operation (>40%) at low power (below 1 kW) and high propellant throughput (more than 100 kg Xe) [6–19]; an overview of several commercially available low-power flight thrusters can be found in [20]. Recent commercial SmallSat constellation concepts have spurred the creation of new start-up companies such as ExoTerra Resource and Apollo Fusion, focused on low-power Hall thruster development. Although performance data on each thruster is presently limited to several plots available on the companies' websites, the performance numbers are promising [21,22]. In general, it appears that new thruster development companies are currently target low-Earth orbit (LEO) spacecraft with low-to-moderate propellant throughput and lifetime requirements (sub-30 kg Xe and sub-4000 h per thruster). While sufficient for LEO missions, these lifetime targets are insufficient for deep-space missions, especially those extending to the outer solar system.

To meet the need of next-generation interplanetary SmallSats, a high-capability low-power EP system is required. The ASTRAEUS (***A****scendant* ***S****ub-kW* ***Tra****nscelestial* ***E****lectric Prop*__*u*__*lsion* ***S****ystem*) program aims to meet the need of deep-space SmallSats by developing a low-power high-efficiency Hall thruster system capable of >100 kg Xe throughput. ASTRAEUS is based around JPL's patent-pending MaSMi (***Ma****gnetically* ***S****hielded* ***Mi****niature*) Hall thruster. MaSMi employs a magnetically shielding magnetic field topology which effectively eliminates ion bombardment erosion of the discharge channel thereby significantly extending the useful life of the device [23–25]. The first generation MaSMi Hall thruster built in 2012 was the first low-power Hall thruster to demonstrate the scalability of magnetic shielding to the sub-kW regime [26,27]. Through numerous design cycles the key limitations to MaSMi's originally poor performance have been systematically identified and eliminated, resulting in incremental performance improvements that have been well documented throughout the program [20,26–32].

The most recent iteration of MaSMi, the development model MaSMi-DM shown in Figure 2, integrates numerous state-of-the-art features including a unique centrally-mounted heaterless hollow cathode called MaSMi's LUC (***L****ow-current* ***U****ltra-compact hollow* ***C****athode*) and a propellant distributor with exceptional azimuthal propellant flow uniformity [33]. Experimentally validated plasma modeling of the MaSMi-DM suggests a throughput capability of >100 kg Xe [33]. To date, the MaSMi-DM has demonstrated a power throttling range of 150 – 1000 W, a peak total thrust efficiency of 53% with $\geq$40% available at >300 W, and a peak total specific impulse of 1940 s with >1500 s available at $\geq$500 W. Preliminary throttling tables derived from the MaSMi-DM's demonstrated performance curves are presented below for the following mission trajectory analysis; a thorough review of the thruster's performance will be available in a future journal publication.

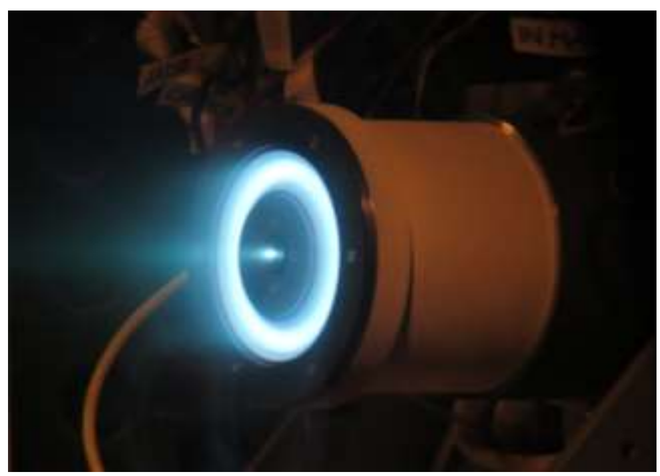

Figure 2. Operation of the MaSMi-DM Hall thruster at the JPL High Bay vacuum facility.

4. Notional Spacecraft Architecture

The initial mass target for the OS^3 spacecraft using the combined propulsion system was 300 kg, making it adaptable to the ESPA GRANDE launch vehicle secondary payload adapter. While the dry mass of the spacecraft remained at approximately 50% of this value, the propellant mass required to complete outer solar system missions drove the total wet mass to 400 kg.

Table 1. Mass Estimates for the OS^3 Spacecraft for Saturn Missions

Subsystem	Mass (kg)	% Dry Mass	% Wet Mass
Payload	22	14%	-
Power			
RTG	60	37%	-
Power Electronics	1	1%	-
Command & Data Handling	1	1%	-
Attitude Control	7	4%	-
Thermal	1	1%	-
Telecom	15	9%	-
Propulsion			
DeSAPS	10	6%	-
EP System Tankage	10	6%	-
Hybrd Rocket	10	6%	-
Hybrid Tankage	5	3%	-
Structures	20	12%	-
CBE Total Mass	**162**	**100%**	**41%**
Margin (25%)	26	-	7%
Propellant			
Xenon	180	-	45%
PMMA	32	-	8%
Total Wet Mass	**400**	**-**	**100%**

Table 1 presents a mass estimate for the OS^3 spacecraft. A 22 kg scientific payload is assumed. Subsystems are assumed to be cubesat/SmallSat-scale systems with the exception of telecom and power. The telecom subsystem mass was increased to allow for a deployable 1.5 m high gain antenna and the power subsystem mass was increased to allow for an RTG. The RTG was assumed to be a NextGen RTG (also referred to as the Advanced RTG[1]) that would provide at least 390 W at the end of its 17 year operating life. The system margin is calculated against the spacecraft CBE mass less the RTG mass. It is assumed that the 60 kg mass estimate for the RTG is a maximum mass allocation for the RTG.

5. Mission Trajectory Studies

Trajectory Modeling Tools

The JPL in-house low thrust trajectory solver called the Mission Analysis Low-Thrust Optimizer (MALTO) was used to generate the mission trajectories presented in this section. MALTO is well-documented and widely used. Details about this code and its applications can be found in [35].

Propulsion Subsystem Assumptions

In order to produce the mission trajectories discussed herein, several assumptions must be made for both elements of the propulsion subsystem.

Chemical Propulsion Element: The chemical system would primarily be used for high thrust burns inside of a gravity well such as either Earth escape or orbit insertion. For these maneuvers, the trajectory design assumes a specific impulse of 311 seconds. For deep space maneuvers, electric propulsion would be used.

Electric Propulsion Element: It is assumed that a flight version of the MaSMi Hall thruster is capable of a propellant throughput of 200 kg Xe, regardless of throttling condition. This assumption is supported by two previous experimental and computational investigations completed in 2015 and 2017. The 2015 effort suggested that the primary wear mechanisms in the MaSMi-60-LM1 (specifically, energetic ion sputter erosion of the discharge channel) had been reduced by a factor of at least 10 – 100 compared to conventional unshielded Hall thrusters [31,32]. The 2017 effort's experimentally validated plasma simulations estimated a throughput capability of >>100 kg Xe, as mentioned above [33]. This assumption is consistent with previous mission studies with the MaSMi thruster [20].
For the purposes of the mission analysis, the mission trajectories were generated using the as-demonstrated performance of the MaSMi-DM thruster. Power was assumed to be delivered to the thruster via a power processing unit (PPU) with a constant 96% efficiency across all thruster operating conditions. An additional 5% margin and 10% contingency on the EP system's required power were also applied. In equation form, the electric propulsion system power ($P_{EP\ System}$) can be expressed as

$$P_{EP\ System} = 1.1 \times \left[1.05 \times \left(\frac{P_d}{\eta_{PPU}}\right)\right] \qquad (1)$$

where P_d is the thruster discharge power and η_{PPU} is the PPU efficiency. The thruster throttling tables presented in Figure 3 use the EP system power calculated from Equation 1 to provide a conservative estimate of the system-level performance.

For the missions below, ASTRAEUS would be operated at 300 W input power at a 90% duty cycle, resulting in an Isp of 1164 s and a thrust of 17.5 mN based on the throttling tables above. The hybrid rocket would be used for orbit insertion burns at each planet.

Mission Target 1: Saturn

Figure 4 shows a trajectory that reaches Saturn in 7.5 years with a launch hyperbolic escape velocity (C3) of 25 km^2/s^2 (dedicated launch vehicle). This trajectory only uses Earth flybys, and as such would make launch opportunities recur every year. The heliocentric arrival velocity (v_∞) at Saturn is only 3.5 km/s, which allows for a modest 263 m/s Saturn orbit insertion burn for 180 day initial orbit. This corresponds to 22 kg of hybrid rocket PMMA propellant.

After capture at Saturn, the OS^3 spacecraft would have a delivered mass of more than 180 kg (45% of launch mass), including the remaining propellant. This includes 10 kg of PMMA and 21 kg of Xe, corresponding to about 140 m/s of ΔV for the hybrid rocket and >1 km/s of ΔV for ASTRAEUS. This ΔV capability should be sufficient for a Titan orbiter,

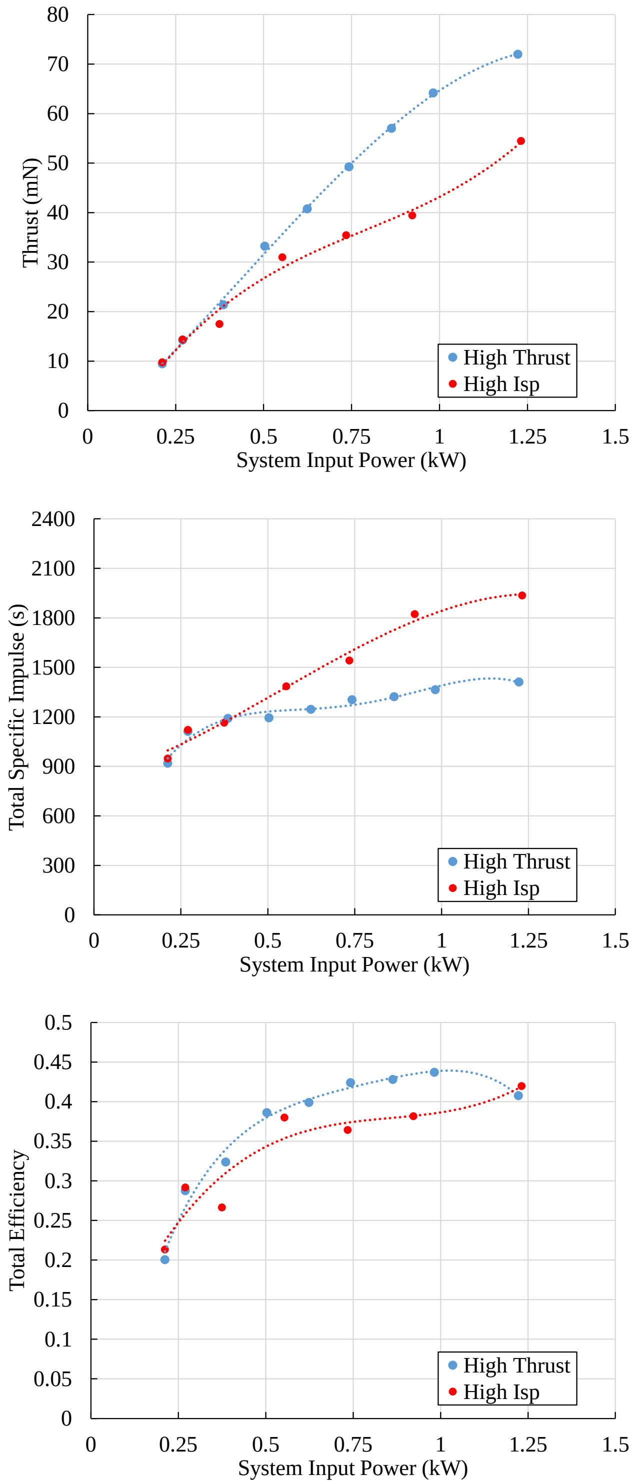

Figure 3. Throttling tables for the electric propulsion element using the as-demonstrated performance of the MaSMi-DM thruster with an assumed 96% efficient PPU, a 5% power margin, and a 10% power contingency.

Enceladus orbiter, or Enceladus lander mission [35,36].

Mission Target 2: Uranus

Figure 5 shows a trajectory that reaches Uranus in 13.5 years with a launch C3 of 107 km^2/s^2 (dedicated launch vehicle, possibly with SRM kick stage). This trajectory requires a Jupiter flyby, forcing the launch opportunities to be phased with the Jovian orbit.

The EP system is used to reduce the Uranus arrival velocity to 3 km/s. This enables orbit capture into a 180 day orbit with a 266 m/s burn. The interplanetary trajectory require 97 kg of Xe propellant and the capture while the hybrid rocket requires 26 kg of PMMA propellant. This results in a wet mass of 276 kg at Uranus with 83 kg of Xenon. This would allow 4 km/s of ΔV in the Uranian system, which is more than sufficient for an ambitious gravity-assist tour with low-speed flybys of the Uranian moons.

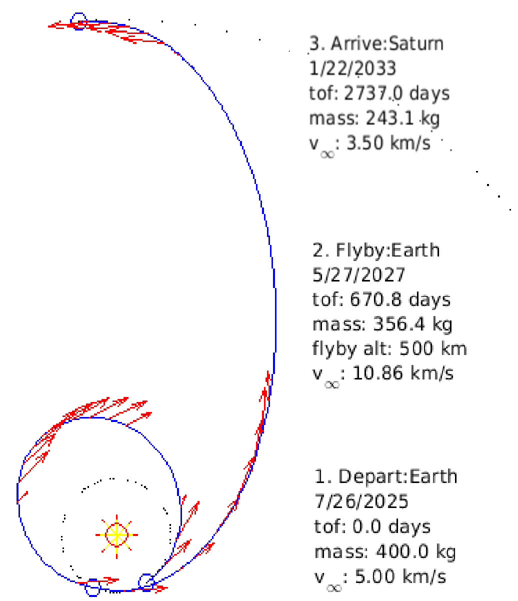

Figure 4. Example 7.5 year trajectory to Saturn using the notional OS³ spacecraft.

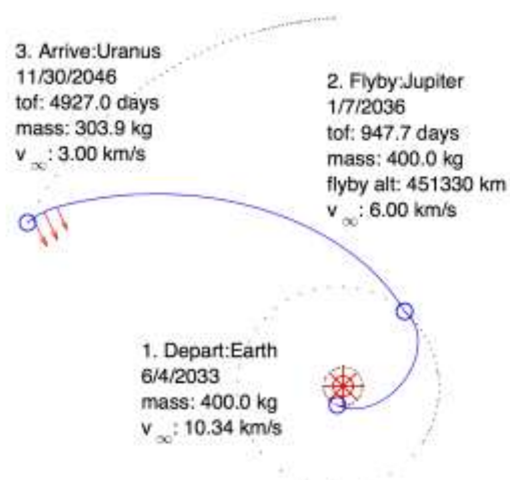

Figure 5. Example 13.5 year trajectory to Uranus using the notional OS³ spacecraft.

6. Conclusion

Recent JPL developments in SmallSat chemical and electric propulsion technologies were summarized and applied to a notional outer solar system SmallSat spacecraft. The hybrid propulsion system has been demonstrated under relevant conditions (TRL 5) and has shown stable, high performance and reliable ignition. The key design features and performance of the MaSMi-DM Hall thruster, a future component of the low-power ASTRAEUS electric propulsion system, were described. Preliminary throttling curves based on the as-tested performance of the MaSMi-DM thruster were presented and used for mission trajectory analysis.

A notional SmallSat spacecraft architecture, limited to 400 kg wet mass and using a combined chemical & electric propulsion system, was developed for mission concepts to Saturn and Uranus. In both cases, a Next-Generation RTG providing at least 390 W of power at end of life was assumed as the spacecraft power source. The example mission trajectory to Saturn delivered more than 240 kg into the Saturnian system in 7.5 years with enough remaining propellant for both the hybrid rocket and electric propulsion system to enable follow-on missions to Titan and Enceladus. The Uranus mission trajectory required a longer 13.5 year flight time, but suggested the possibility of delivering more than 275 kg into the Uranian system – a feat that, to date, has never been accomplished. The two mission concepts presented clearly demonstrate the mission-enabling capabilities of a combined hybrid rocket and long-life electric propulsion system to SmallSat-sized spacecraft.

7. Future Work

The MaSMi-DM Hall thruster will be advanced to an engineering model design, which is expected to have the same strong performance as demonstrated by the development model. The MaSMi-EM will undergo long-duration wear testing and dynamic & thermal environments testing in the near future. The remaining subsystem components of ASTRAEUS will also be developed, assembled, and tested. Performance and characterization testing of the flight-like hybrid rocket motor will also continue. As updated information becomes available on either the hybrid or electric propulsion elements discussed above, additional mission studies will be performed to show the enabling features of a combined chemical & electric propulsion system.

Acknowledgements

This research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. The information presented about future SmallSat mission concepts is pre-decisional and is provided for planning and discussion purposes only.

References

[1] "NASA Nuclear Power Assessment Study: Final Report," TSSD-23122, JHU/APL (2015).

[2] Gibson, M. A., Oleson, S. R., Poston, D. I., and Mcclure, P., "NASA ' s Kilopower Reactor Development and the Path to Higher Power Missions," *IEEE Aerospace Conference*, Big Sky, MT, Mar. 2017.

[3] Jens, E. T., Karp, A. C., Rabinovitch, J., Nakazono, B., Conte, A., and Vaughan, D., "Design of Interplanetary Hybrid CubeSat and SmallSat Propulsion Systems," AIAA-2018-4668, *54th AIAA Joint Propulsion Conference*, Cincinnati, OH, Jul. 2018.

[4] Rabinovitch, J., Jens, E. T., Karp, A. C., Nakazono, B., Conte, A., and Vaughan, D. A., "Characterization of PolyMethylMethAcrylate as a Fuel for Hybrid Rocket Motors," AIAA-2018-4530, *54th AIAA Joint Propulsion Conference*, Cincinnati, OH, Jul. 2018.

[5] Conte, A., Rabinovitch, J., Jens, E., Chandler Karp, A., Nakazono, B., and Vaughan, D., "Design, Modeling and Testing of a O2/CH4 Igniter for a Hybrid Rocket Motor," AIAA-2018-3239, *Fluid Dynamics Conference, AIAA Aviation Forum*, Atlanta, GA, Jun. 2018.

[6] Galitsky, S., "Electric Propulsion in Russia," *News from Moscow*, vol. 26, , pp. 9–13 (2000).

[7] Kim, V., Popov, G., Arkhipov, B., Murashko, V., Gorshkov, O., Koroteyev, A., Garkusha, V., Semenkin, A., and Tverdokhlebov, S., "Electric Propulsion Activity in Russia," IEPC-01-05, *27th International Electric Propulsion Conference*, Pasadena, CA, Oct. 2001.

[8] Kim, V., "Stationary Plasma Thrusters in Russia : Problems and Perspectives," *J. Tr. MAI (Moscow Aviat. Institute)*, vol. 60, (2014).

[9] Guerrini, G., Michaut, C., Vesselovzorov, A. N., and Bacal, M., "Characterization of Plasma Inside the SPT-50 Channel by Electrostatic Probes," IEPC-97-053, *25th International Electric Propulsion Conference*, Cleveland, OH, Aug. 1997.

[10] Murashko, V. M., Koryakin, A. I., Nyatin, A., Kim, V., Popov, G., Romashko, A., Yermoshkin, Y., Petrusevich, V., Gorshkov, O., Koroteyev, A., Garkusha, V., Semenkin, A., and Tverdokhlebov, S., "State of the Art and Prospects of Electric Propulsion in Russia," IEPC-2003-340, *28th International Electric Propulsion Conference*, Toulouse, France, Mar. 2003.

[11] Garner, C. E., Polk, J. E., Pless, L. C., Goodfellow, K. D., and Brophy, J. R., "Performance Evaluation and Life Testing of the SPT-100," IEPC-93-091, *23rd International Electric Propulsion Conference*, Seattle, WA, 1993.

[12] Garner, C. E., Brophy, J. R., Polk, J. E., and Pless, L. C., "A 5,730-Hr Cyclic Endurance Test Of The SPT-100," AIAA-95-2667, *31st AIAA Joint Propulsion*

Conference, San Diego, CA, Jul. 1995.

[13] Gnizdor, R., Kozubsky, K., Koryakin, A., Maslennikov, N., Pridannikov, S., and Day, M., "SPT100 Life Test with Single Cathode Up to Total Impulse Two Million Nsec," AIAA-98-3790, *34th AIAA Joint Propulsion Conference*, Cleveland, OH, 1998.

[14] Ragot, X. and Mattei, N., "Plasma Propulsion Sub-System Life Test - ALCATEL," ESA SP-465, *3rd International Conference on Spacecraft Propulsion2*, Cannes, France, Oct. 2000.

[15] Hruby, V., Monheiser, J., Pote, B., Kolencik, J., Freeman, C., and Rostler, P., "Development of Low Power Hall Thrusters," AIAA-99-3534, *30th Plasmadynamics and Lasers Conference*, Norfolk, VA, Jul. 1999.

[16] BUSEK, "BHT-200 Hall Thruster." 2013.

[17] Beal, B. E., Gallimore, A. D., and Hargus, W. A., "Preliminary Plume Characterization of a Low-Power Hall Thruster Cluster," AIAA-2002-4251, *38th AIAA Joint Propulsion Conference*, Indianapolis, IN, 2002.

[18] Monheiser, J., Hruby, V., Freeman, C., Connolly, W., and Pote, B., "Development and Testing of a Low-Power Hall Thruster System," *Micropropuls. Small Spacecr.*, vol. 187, , pp. 255–270 (2000).

[19] Hargus, W. A. and Charles, C. S., "Near Exit Plane Velocity Field of a 200 W Hall Thruster," AIAA-2003-5154, *39th AIAA Joint Propulsion Conference*, Huntsville, AL, Jul. 2003.

[20] Conversano, R. W., Goebel, D. M., Hofer, R. R., and Arora, N., "Performance Enhancement of a Long-Life , Low-Power Hall Thruster for Deep-Space SmallSats," *IEEE Aerospace Conference*, Big Sky, MT, Mar. 2017.

[21] ExoTerra Resource, "Halo Micro Hall Effect Thruster." https://exoterracorp.com/?p=114

[22] Apollo Fusion, "Apollo Constellation Engine." http://apollofusion.com/index.html.

[23] Mikellides, I. G., Katz, I., Hofer, R. R., Goebel, D. M., de Grys, K., and Mathers, A., "Magnetic shielding of the channel walls in a Hall plasma accelerator," *Phys. Plasmas*, vol. 18, no. 033501 (2011).

[24] Mikellides, I. G., Katz, I., Hofer, R. R., and Goebel, D. M., "Magnetic shielding of a laboratory Hall thruster. I. Theory and validation," *J. Appl. Phys.*, vol. 115, no. 043303 (2014).

[25] Hofer, R. R., Goebel, D. M., Mikellides, I. G., and Katz, I., "Magnetic shielding of a laboratory Hall thruster. II. Experiments," *J. Appl. Phys.*, vol. 115, no. 043303 (2014).

[26] Conversano, R. W., Goebel, D. M., Hofer, R. R., Matlock, T. S., and Wirz, R. E., "Magnetically Shielded Miniature Hall Thruster : Development and Initial Testing," IEPC-2013-201, *33rd ERPS International Electric Propulsion Conference*, Washington D.C., Oct. 2013.

[27] Conversano, R. W., Goebel, D. M., Hofer, R. R., Matlock, T. S., and Wirz, R. E., "Development and Initial Testing of a Magnetically Shielded Miniature Hall Thruster," *IEEE Trans. Plasma Sci.*, vol. 43, no. 1 (2015).

[28] Conversano, R. W., Goebel, D. M., Mikellides, I. G., Hofer, R. R., Matlock, T. S., and Wirz, R. E., "Magnetically Shielded Miniature Hall Thruster: Performance Assessment and Status Update," AIAA-2014-3896, *50th AIAA Joint Propulsion Conference*, Cleveland, OH, Jul. 2014.

[29] Conversano, R. W., Goebel, D. M., Hofer, R. R., Mikellides, I. G., Katz, I., and Wirz, R. E., "Development and Validation Overview for the Design of Low Power Magnetically Shielded Hall Thrusters," *62nd JANNAF Propulsion Meeting*, Nashville, TN, Jun. 2015.

[30] Conversano, R. W., Goebel, D. M., Hofer, R. R., Mikellides, I. G., Katz, I., and Wirz, R. E., "Magnetically Shielded Miniature Hall Thruster: Design Improvement and Performance Analysis," IEPC-2015-100 / ISTS-2015-b-100, *34th ERPS International Electric Propulsion Conference*, Kobe, Japan, Jul. 2015.

[31] Conversano, R. W., Goebel, D. M., Hofer, R. R., Mikellides, I. G., and Wirz, R. E., "Performance Analysis of a Low-Power Magnetically Shielded Hall Thruster: Experiments," *AIAA J. Propuls. Power*, vol. 33, no. 4, pp. 975–983 (2017).

[32] Conversano, R. W., Goebel, D. M., Mikellides, I. G., Hofer, R. R., and Wirz, R. E., "Performance Analysis of a Low-Power Magnetically Shielded Hall Thruster: Computational Modeling," *AIAA J. Propuls. Power*, vol. 33, no. 4, pp. 992–1001 (2017).

[33] Conversano, R. W., Lobbia, R. B., Tilley, K. C., Goebel, D. M., Reilly, S., Mikellides, I. G., and Hofer, R. R., "Development and Initial Performance Testing of a Low-Power Magnetically Shielded Hall Thruster with an Internally-Mounted Hollow Cathode," IEPC-2017-64, *35th International Electric Propulsion Conference*, Atlanta, GA, Oct. 2017.

[34] Sims, J., Finlayson, P., Rinderle, E., Vavrina, M., and Kowalkowski, T., "Implementation of a Low-Thrust Trajectory Optimization Algorithm for Preliminary Design," AIAA 2006-6746, *AIAA/AAS Astrodynamics Specialist Conference and Exhibit*, Keystone, CO, Aug. 2006.

[35] Strange, N., Spilker, T., Landau, D., Lam, T., Lyons, D., and Guzman, J., "Mission Design for the Titan Saturn System Mission Concept," AAS-09-356, *AAS/AIAA Astrodynamics Conference*, Pittsburgh, PA, Aug. 2009.

[36] Strange, N. J., Campagnola, S., and Russell, R. P., "Leveraging Flybys of Low Mass Moons to Enable an Enceladus Orbiter," AAS-09-435, *AAS/AIAA Astrodynamics Conference*, Pittsburgh, PA, Aug. 2009.

BIOGRAPHY

Ryan W. Conversano *received a B.S., M.S., and Ph.D. in Aerospace Engineering from the University of California, Los Angeles in 2010, 2011, and 2015 respectively. He is a Technologist in the Electric Propulsion Group at the Jet Propulsion Laboratory where he is responsible for the development and testing of high-efficiency, long-life electric thruster technologies. Dr. Conversano is the PI of the* ***A****scendant* ***S****ub-kW* ***Tra****nscelestial* ***E****lectric Prop****u****lsion* ***S****ystem (ASTRAEUS) and the MaSMi Hall thruster development programs. He has authored more than 20 technical publications in the fields of electric propulsion and plasma physics and is the lead inventor on the pending MaSMi Hall thruster patent.*

Jason Rabinovitch *is a Mechanical Engineer at the Jet Propulsion Laboratory (JPL), California Institute of Technology, where he works in the Entry, Descent, and Landing & Formulation Group. Prior to JPL, Dr. Rabinovitch received a B.Sc. in Mechanical Engineering from Yale University in 2008, a M.Sc. in Aerospace Engineering from the California Institute of Technology in 2009, a M.Sc. in Fluid Mechanics from École Polytechnique (Paris) in 2010, and a Ph.D. in Aeronautics from Caltech in 2014. Dr. Rabinovitch is a member of the AIAA Thermophysics Technical Committee, and his research interests span a wide range of topics related to experimental and computational fluid mechanics applied to EDL, vehicle design, propulsion, ablation, and geophysical applications.*

Nathan Strange *has a B.S. degree in Physics from Purdue University as well as a B.S., M.S., and a Ph.D. in Aeronautics & Astronautics from Purdue University. He has worked at NASA's Jet Propulsion Laboratory since 2000, where he currently the Supervisor for the Mission Engineering and Planning Group. Previously, he was the Mission Design Manager for the Asteroid Redirection Robotic Mission. He has also worked on the Navigation Team for the Cassini-Huygens mission to Saturn and has supported the development of many robotic and human interplanetary mission concepts as both a mission designer and a systems engineer.*

Nitin Arora *is a systems engineer in the Mission Concept Formulation Group at JPL. His interests lie in the field of mission design, spacecraft system engineering, trajectory optimization, gravity field computation and high-performance computing. Nitin received his Ph.D. in Aerospace engineering from Georgia Institute of Technology in 2013, his Masters in Aerospace engineering from Georgia Institute of Technology in 2009, and his Bachelors of Technology from Punjab Engineering College, India, in 2007.*

Elizabeth Jens *received a Ph.D. in Aeronautics and Astronautics from Stanford University in 2016. Dr. Jens was recognized as one of Australia's Most Innovative Engineers of 2017 as Determined by Engineers Australia. She has twice been an Amelia Earhart Fellow (2012 and 2014), was a Rotary Ambassadorial Scholar (2010) and the Australian Fulbright Scholar in Science and Engineering (2010). She received a bachelor of science in physics and a bachelor of mechanical engineering with first class honors from the University of Melbourne, Australia in 2008. Dr. Jens is currently a Propulsion and Systems Engineer at the NASA Jet Propulsion Laboratory. She splits her time between working on a cold gas system for the Mars 2020 mission and developing a hybrid propulsion system for small-scale interplanetary missions. She is a member of the Propulsion Technical Committee of the International Astronautical Federation and the Hybrid Rocket Technical Committee of the American Institute of Aeronautics and Astronautics.*

Ashley C. Karp *is the Propulsion Lead and JPL Deputy Manager for the Mars Ascent Vehicle Study. She is also the PI for JPL's Hybrid Propulsion Test Facility. She is the Chair of the AIAA Hybrid Rocket Propulsion Technical Committee. She earned a Ph.D. in Aeronautics and Astronautics from Stanford University in 2012 and a B.A. in Astrophysics, Physics and Political Science from the University of California, Berkeley in 2005. She has worked on the Mars 2020 propulsion system and has been involved with many mission concept studies.*